PRELIMINARY

DESIGN

of

HEAVY

BOMBER

with

4 P+W TWIN WASP MOTORS

4000 H.P.

Model S-12

Designer ~ W.F. Schult

Engineer ~ W.F. Schult

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ANALYSIS General
ENGINEER Wm F. Schult
DATE 11-1-40

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DESCRIPTION

In this design the main feature is to investigate weather an airplane with medium wing loading (25#/□) and no flaps, would have a comparative performance with the best of other types having a relatively high wing loading (40#/□) and flaps necessary for low landing speed.

For comparison, I chose two recent airplanes, namely Lockheeds new 4 motored transport and Consolidated's new 4 motored army bomber. Since these ships have about a 40#/□ wing loading they necessarily have to have flaps to get a minimum speed of 80 m.p.h.

In order to accomplish what I started out to do it seemed necessary to design a flying wing. Since with this type a lower wing loading is possible with-out the use of excessive wing area, because practically all of the structure contributes to lift, it was considered tentively. On the other hand the general opinion is a little skeptical about flying wings as yet. Therefore I thought of a compromise in using a main wing of relatively high $C_{L\,max}$ and low $C_{D\,min}$ with a small C_{m_o}. This wing is then increased in depth at the center sufficient to house the bombs, extended forward to make room for the personnel and extended rearward only sufficiently to mount a small elevator. The rudders being mounted on the wing trailing edge and rather near the tips. Furthermore, it seemed advisable to place the motors in the wing and mount the propellers in the rear since this installation will probably be the most efficient. A tricycle landing gear is used for best all around design.

The next consideration is chose of wing section. I selected the N.A.C.A. 24 since it has a relatively high $C_{L\,max}$ (1.67) and a high $C_{L\,max}/C_{D\,min}$ (246) and a zero C_{m_o}.

After considerable calculation I decided on using the above section with 10% thickness, aspect ratio of 10 and a .3 taper ratio. The results indicate a $C_{L\,max}$ (1.54) at 80 m.p.h., a $C_{D\,min}$ (.0058) at 315 m.p.h. and C_{m_o} is considered zero.

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DESCRIPTION

Since the wing C_{mo} is practically zero the elevator area calculated for static stability is relatively small (160 □'). The rudders being mounted on the wing instead of on the tail thus allowing more freedom for manuvering of the rear guns, so essential in a bomber. Although this rudder arraingement is not so effective at small angles of motion, it is ample at 20° travel).

The resulting preliminary design is estimated to weigh about 40000# with a bomb load of 8800# and 1000# of ammunition for 4-.50 caliber machine guns. Its personnel consists of 6 men, 1 commander & radio operater combined, 2 pilots, 1 bomber and 2 gunners.

For comparative performance see table below

	LOCKHEED	CONSOLIDATED	SCHULT S-12
Weight #	40000	40000	39964
H.P.	4000	4000	4000
W/S #	40	Data Not Available Yet	25
V_{min} S.L. m.p.h	79		80
V_{max} S.L. m.p.h.	300		315
R/C S.L. l/min	1515		1900
Service Ceiling'	25200		31500

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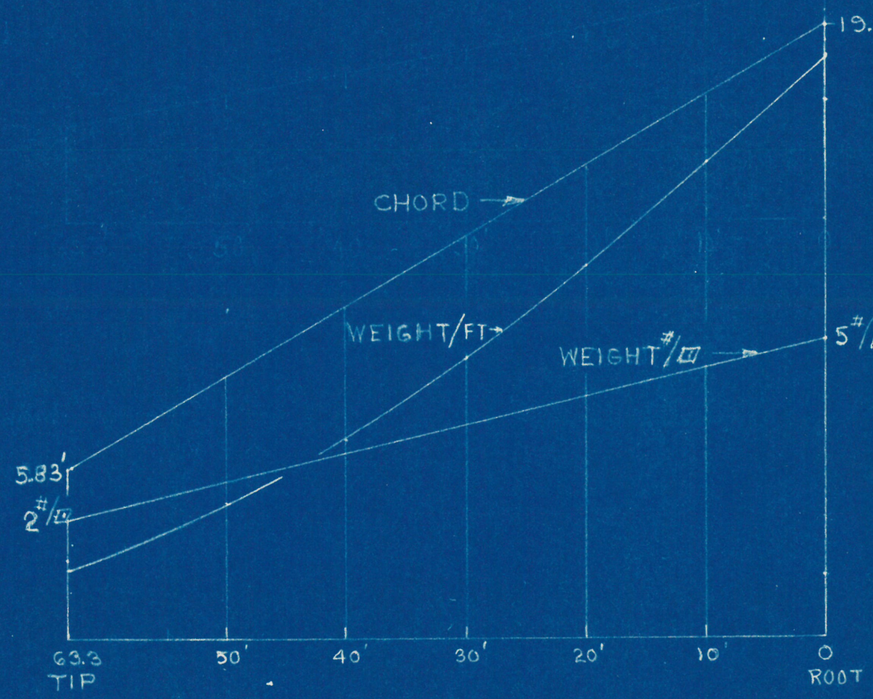
WING WEIGHT
ESTIMATE

Tip Weight = 2 #/□

Root " = 5 #/□



①	2	2.63	3.11	3.67	4.15	4.62	5	Weight #/□
②	5.83	8.70	10.86	13.0	15.16	17.32	19.5	Chord '
③	11.66	22.9	33.8	47.7	62.8	80	97.5	Weight #/ Ft.
④	≤③=356.36	356.36/7 × 63.3/800 = 4 #						Weight #/□ Ave.
⑤	738.	1145	1352	1425	1256	800	0	Moment '#
⑥	≤M = 6712 '#	6712/356.36 = 18.8' from Root						C.G.

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~ WEIGHT ~

STRUCTURE

NO.	ITEM	ESTIMATE	ACTUAL
I	WING (4#/□ ×1600)	6400.	
	TOTAL	6400	6400
	% W_s	55	
II	CENTER BODY		
	Structure	1600.	
	Enclosure, Pilots & Commander	175.	
	Enclosure, Bomber	120	
	Wind Brake (Retractable) Top Gunner	30	
	Trap Door (Lower Gunner)	25	
	TOTAL	1950.	1950.
	% W_s	16.75	
III	LANDING GEAR (NOSE)		
	Oleo	150.	
	Cowling	10.	
	Wheel (1-45" Dia) & Tire	185.	
	Structure & Raizing Mech.	130.	
	TOTAL	475.	475
	% W_s	4.07	
IV	LANDING GEAR (REAR)		
	Oleo	500.	
	Cowling	30.	
	Wheel, Tire & Brake (2-56" Dia.) 2x370	740	
	Structure & Raizing Mech.	300.	
		1570	1570
	% W_s	13.5	
V	NACELLE (Structure) 175 x 4	700.	700.
	% W_s	6.01	

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WEIGHT

STRUCTURE

NO.	ITEM		ESTIMATE	ACTUAL
VI	TAIL			
	Horizontal (1600 x .10)	160□ ~ 160 x 2#	320	
	Vertical (1600 x .09)	150□ 150 x 1.5#	225.	
		TOTAL	545.	545
		% W_s	4.67	
	STRUCTURE =		11,640	
		% W_e	50.6	

POWER PLANT

VII	ENGINE			
	P+W. Twin Wasp S3C4-G			
	(1000 H.P. ~ 2700 R.P.M. 14500' Alti; 100 Octane)			
		1460# x 4	5840.	
	Accessories	67# x 4	268.	
	Starter		160.	
a	Generator		35.	
	Battery		140.	
	Cables		40.	
	Controls		125.	
	Cooling System	25 x 4	100.	
b	Propeller	350 x 4 (11' Dia.)	1400.	
c	Exhaust System	(1000 x .05 x 4)#	200.	
d	Shafting	(2" O.D. x .125 x 180") ~ 340 x .112# x 2	76.	
		(2" O.D. x .125 x 160") ~		
e	Coupling	75 x 4	300.	
f	Bearing	30 x 4	120.	
		TOTAL	8804.	8,804.
		% W_e P.P.	91.1	

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WEIGHT

POWER PLANT

NO.	ITEM	ESTIMATE	ACTUAL
VIII	FUEL SYSTEM (800 Gals		
	Tank (Integral Wing) 50#x4	200.	
	Piping 20#x4	100.	
	TOTAL	300.	300.
	% P.P.	3.1	
IX	OIL SYSTEM (800 x .09) = 72 Gals		
	Tank 72 x 1.5	108.	
	Radiator 100#x4	400.	
	Piping	40.	
	TOTAL	548	548
	% P. P.	5.8	
	POWER PLANT	9652	
	% We	42.	

FIXED EQUIPMENT

X	INSTRUMENTS		
	Flight, Motor, Landing etc	120.	120
	% Fix. Eq.	7.05	
XI	SURFACE CONTROLS		
	Elevator Control Column	40.	
	Rudder Pedals	25.	
	Elevator & Rudder Cables	45.	
	Aileron Cables	30.	
	Trim Tab Cables (Elev, Rudder, Ail.)	50.	
	TOTAL	190.	190
	% Fix. Eq.	11.2	
XII	FURNISHINGS		
	Seats 16#x6	96.	
	Floor & Walls (Piots & Commander)	75.	
	Radio Operators Table etc	35	
	Safety Belts 4#x6	24.	

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2

WEIGHT

FIXED EQUIPMENT

NO.	ITEM		ESTIMATE	ACTUAL
XII	FURNISHINGS			
	Instrument Panel		45.	
	Fire Extinguisher 15# x 6		90.	
	Floor (Rear Gunner, Top)		25.	
		TOTAL	390.	390
		% Fix. Eq.	22.9	
XIII	ELECTRICAL EQUIPMENT			
		% Fix. Eq.	11.75	200
XIV	RADIO EQUIPMENT			
	Radio (Receiving + Telephone)		100.	
	Interphone		40.	
	Telegraph		50.	
		TOTAL	190.	190.
		% Fix. Eq.	11.2	
XV	ARMAMENT INSTALLATION			
	Gun Mounts 30# x 4		120.	
	Bomb Racks 50# x 8		400.	
	Bomb Sight Support		25.	
	Ammunition Draws 10# x 4		40.	
	Miscellaneous		25.	
		TOTAL	610.	610.
		% Fix. Eq.	35.9	
		FIXED EQUIPMENT		1700.
		% We	7.4	
		WEIGHT EMPTY		22992
		% Wg	57.5	

USEFUL LOAD

XVI	EQUIPMENT (Special)			
	Signal		25.	
	Oxygen 22# x 6		132.	
	Flares 33# x 4		132.	
	Gunnes (50 Cal.) 52# x 4		208.	
	Bomb Sight		15.	
		TOTAL	512.	512
		% Use Ld.	2.93	

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WEIGHT

USEFULL LOAD

NO	ITEM	ESTIMATE	ACTUAL
XVII	BOMBS & AMMUNITION		
	Bombs 1100# x 8	8800.	
	Ammunition (4000 Rds) .25 x 1000 x 4	1000.	
	TOTAL	9,800.	9800.
	% Use. Ld.	58	
XVIII	PERSONNEL		
	Commander (Radio Operator)	200.	
	Pilots 200 x 2	400.	
	Bomber (Gunner Front)	200.	
	Gunner (Rear Upper)	200.	
	" (Rear Lower)	200.	
	TOTAL	1200	1200.
	% Use. Ld.	7.1	
XIX	FUEL & OIL		
	Fuel .6# / H.P.HR. 4-1000 H.P.		
	.6 x 1000 x 4 x 2 Hours		
	4800/6 = 800 Gallons	4800.	
	Oil (.11 x 800) = 88 Gal's		
	88 x 7.5	660.	
	TOTAL	5460.	5460
	% Use. Ld.	32.4	
	USEFULL LOAD		16972.
	% W_g	42.5	

GROSS WEIGHT 39964

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ITEM	UNIT	WEIGHT #	HORIZONTAL DIST. AFT REF. LINE	HORIZONTAL MOMENT "#	VERTICAL DIST. VERT. REF. LINE	VERTICAL MOMENT "#
I	WING	6400	285	1820,000	130	832,000
II	CENTER BODY	1950	182	354,020	105	205,000
III	LANDING GEAR (NOSE)	475	98	46650	68	32300
IV	LANDING GEAR (REAR)	1570	337	529,900.	96	151,000
V	NACELLE (STRUCTURE)	700	290	203,000.	136	95200
VI	TAIL	545	596	325,000.	90	49100
VII	ENGINE (GROUP)	8804	303	2,678400	126	1110000
VIII	FUEL SYSTEM	300	280	84,000	126	37800
IX	OIL SYSTEM	548	235	129,000	126	69000
X	INSTRUMENTS	120	100	12000	120	14400
XI	SURFACE CONTROLS	190	248	47050	110	20900
XII	FURNISHINGS	390	141	55,110	120	46750
XIII	ELECTRIC EQUIPMENT	200	200	40,000.	130	26000
XIV	RADIO EQUIPMENT	190	160	30,500.	125	23750
XV	ARMAMENT INST.	610	225	137,625	103	63100
XVI	EQUIP. (SPECIAL)	512	137	70,020	106	54225
XVII	BOMBS + AMMUNITION	9800	265	2,604,000	104	1020000
XVIII	PERSONNEL	1200	128'	154,000	107'	129000
XIX	FUEL + OIL	5460	274	1,500,000	126	687,000
XX	GROSS WEIGHT	39,964	x271 =	10,820,325	117	4666,520

REF.	LOADING CONDITION					
①	Empty (Item I to XV)	22,992	282	6,492,305		
②	① + XVI	23504	279	6562,325		
③	② + Personnel	24704	272	6716325		
④	③ + Bombs	33504	271	9,186,325		
⑤	③ + Ammunition	25704	266	6,850,325		
⑥	③ + Bombs + Ammunition	34504	270	9,320,325		
⑦	③ + Fuel + Oil	30,164	272	8,216325		
⑧	⑦ + Bombs + Ammunition	39964	271	10,820,325		
⑨	⑧ Lower Gunner Moved aft + Rudders mounted on Wing	39964	274	10,944,125		

C.P. @ $C_{d_{min}}$ = 25% of M.A.C. = 280" Aft Ref. Line M.A.C. = 167"
C.P. @ $C_{L_{max}}$ = 27% " M.A.C. = 283.3" " " "

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~ WING DIMENSIONS ~

Span = 1520"

Area = 1600 □' - 230000 □"

$Chord_t$ = 5.85' 70.25"

$Chord_s$ = 19.50' 234"

$m_s = m_t$ = 5.80

λ = .3

A.R. = $1520^2/230{,}000$ = 10

$A.R./m_t$ = $10/5.80$ = 1.72

$Y = \frac{(30+40)}{2} \times \frac{1}{760} = .046$

T' = .027

$m' = \frac{5.80}{1+\frac{5.80}{\pi 10}(1+.027)} = 4.87$

$m = m'$ = 4.87

$1/m$ = 0.205

$\alpha_a = \frac{C_L}{m}$

$\alpha^\circ = 57.5 \times \alpha_a$

$= 57.5 \times \frac{C_L}{m}$

$\alpha^\circ = 11.75 \times C_L$

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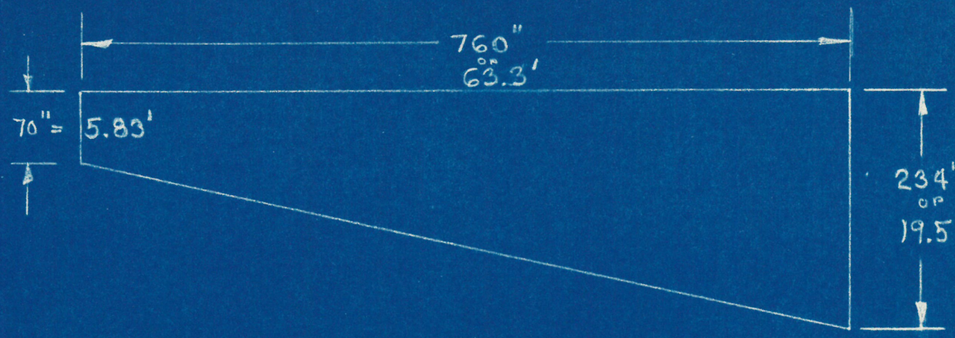

$$m.a.c. = \frac{2}{3}\left(234+70-\frac{234\times70}{234+70}\right) = \frac{2}{3}\times250.2 = 167'' \text{ or } 13.9'$$

$$\text{Aspect Ratio} = \frac{126.6^2}{1600} = 10$$

$$m = \frac{164\,(234+2\times70)}{3\,(234+70} = \frac{164\times374}{3\times304} = 67.2''$$

$$n = 80\ \frac{67.2}{164} = 32.8$$

n_1 (above Ref. Line) = 32.8 + 16.2 + 90 = 139

Location of a.c.

Aft of L.E. of M.A.C. = .25 × 167 = 41.75"

Aft of Ref. Line = 406 − (167 − 41.75) = 280.75"

Above M.A.C. = .06 × 167 × 167 = 10"

Above Ref. Line = 139 + 10 = 149"

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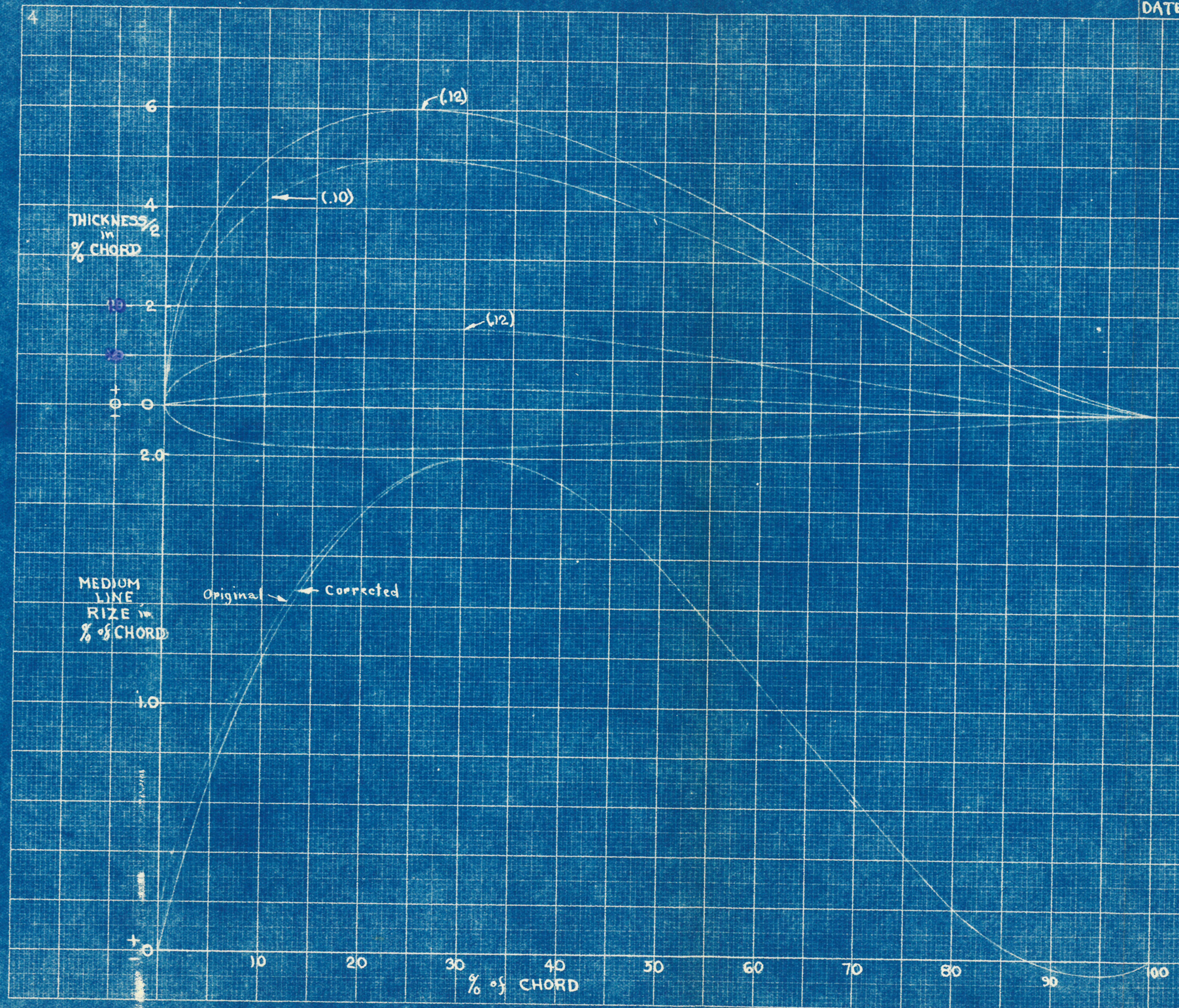

N.A.C.A. 24

Sta.	Medium Line Original	Medium Line Corrected	THICKNESS/2 in % .12	THICKNESS/2 in % .10	THICKNESS/2 in % .10c
1.25	+.395	+.200	2.000	1.670	
2.50	.530	.380	2.800	2.330	
5.00	.795	.690	3.800	3.170	
7.50	1.025	.960	4.480	3.740	
10.00	1.230	1.180	4.970	4.140	
15.00	1.565	1.530	5.595	4.660	
20.00	1.795	1.770	5.900	4.920	
25.00	1.935	1.930	6.000	5.000	
30.00	2.000	2.000	5.920	4.930	4.95
40.00	1.900	1.900	5.460	4.550	
50.00	1.570	1.560	4.620	3.850	
60.00	1.110	1.100	3.620	3.020	
70.00	0.625	.640	2.520	2.100	
80.00	+.205	+.200	1.450	1.210	
90.00	-.035	-.035	.550	0.450	
95.00	-.060	0.060	.210	0.175	
100.00	0.000	0.000	0.000	0.000	

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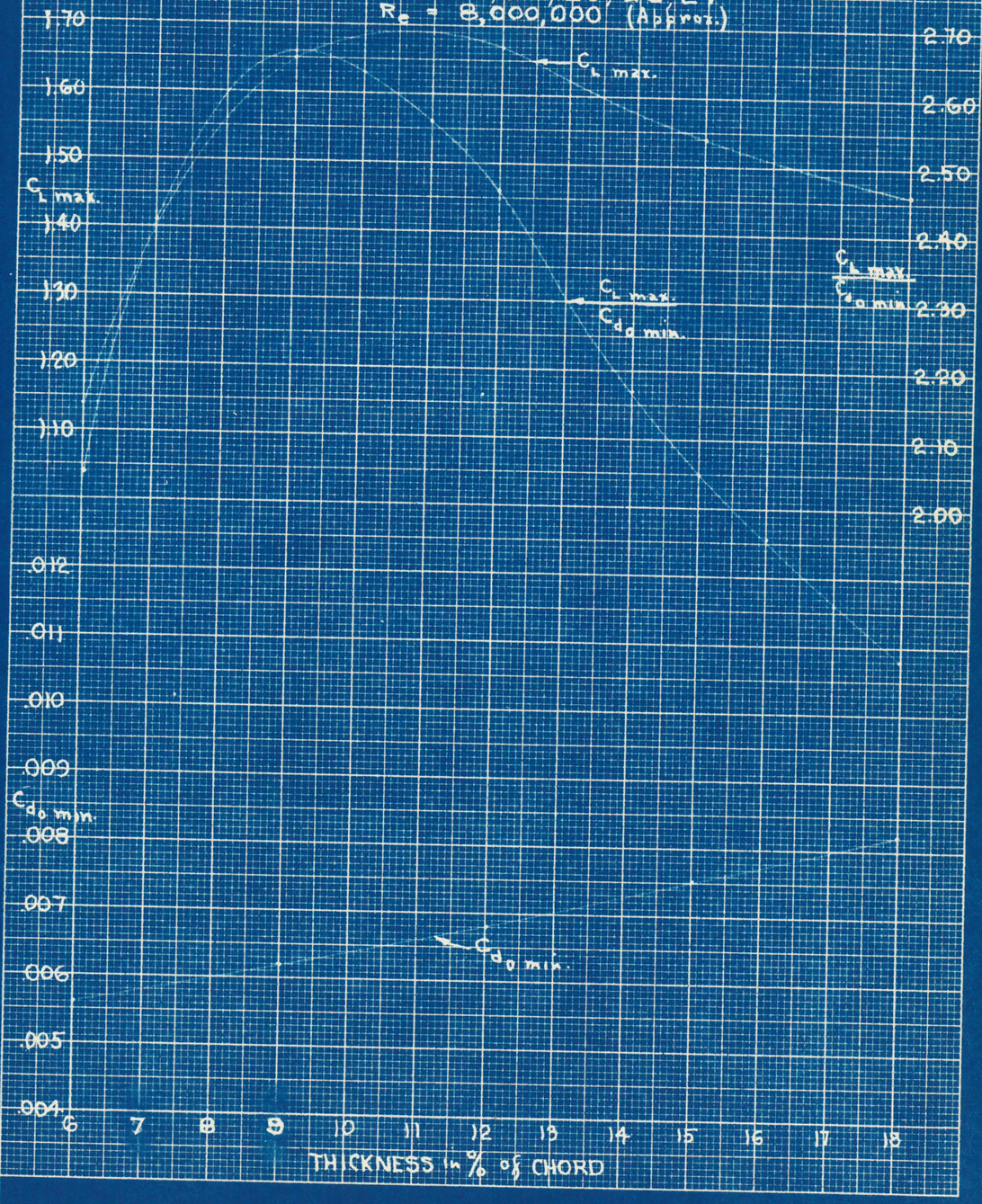
FUNDAMENTAL SECTION CHARACTERISTICS

N.A.C.A. 23, 24, 25, 26, 27

R_e = 8,000,000 (Approx.)



KEUFFEL & ESSER CO., N. Y. NO. 359-11
20 × 20 to the inch, 10th lines heavy.
MADE IN U. S. A.

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.0006
.0004
.0002
ΔC_{d_0}

14
16
20
24
28
32
36
40
27

$\leftarrow .7(\Delta C_{d_0})$ of TABLE III
COMBINATION 186 N.A.C.A.
REPORT NO. 540

INCREASE IN DRAG OF WING IN % DUE TO ENLARGED CENTER PORTION OF WING

16 14 12 10 8 6 4 2 α°

$C_{l\,opt.}$
.20
.10
0
.10

N.A.C.A. - 23, 24, 25, 26

6 8 10 12 14 16 18
THICKNESS in %

$C_l - C_{l\,opt.}$
$C_{l\,max} - C_{l\,opt.}$

0 .1 .2 .3

KEUFFEL & ESSER CO., N. Y. NO. 359-11
20 × 20 to the inch, 10th lines heavy.
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100.
90
80
70
60
50
40
30
20
Re in Millions
10.
9
8
7
6
5
4
3
2
1.

← 24-9 (N.A.C.A.)
N.A.C.A. 24-15 →
N.A.C.A. 23012 → ← N.A.C.A. 24-12
← N.A.C.A. 24-10

.1 +0- .1 .2 .3 .4 .5 .3 .2 .1 0
$\Delta C_{L\,max.}$ C_L

KEUFFEL & ESSER CO., N. Y. NO. 359-60
Semi-Logarithmic, 2 Cycles X 10 divisions.
MADE IN U.S.A.

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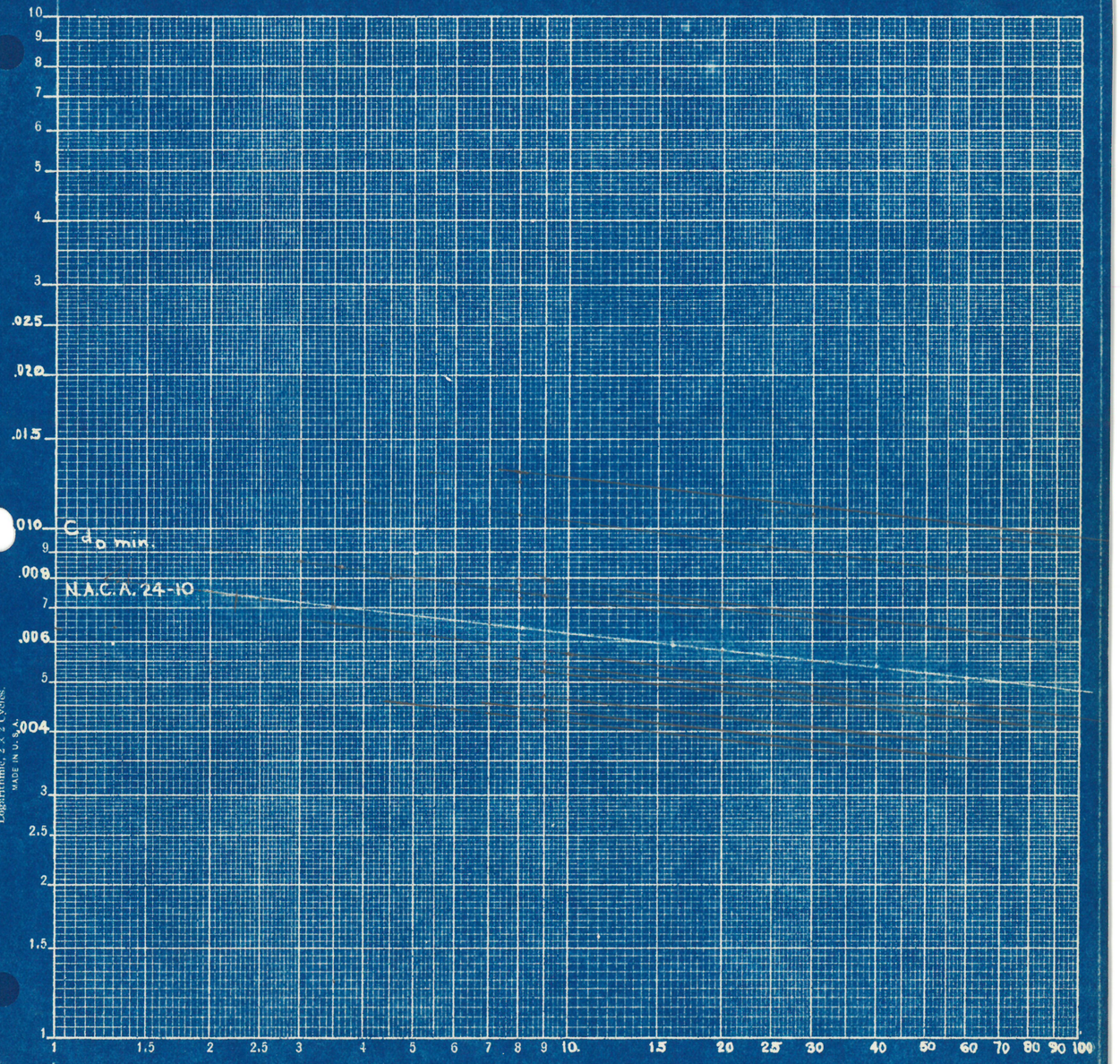

KEUFFEL & ESSER CO., N.Y. NO. 359-110
Logarithmic, 2 x 2 Cycles.
MADE IN U.S.A.

$$C_{d_{o\,min.}} = (C_{d_{o\,min}})_{std.} \left(\frac{(R_e)_{std.}}{R_e} \right)^{0.11}$$

J.S

C_D DETERMINATION of C_L & C_D of COMPLETE AIRPLANE

The wing C_L and C_D coefficients were first calculated for a wing with an aspect ratio of 10 taper ratio of .3 and constant thickness of 10% using the special method (ANC-1 Bulletin). These values were then revised to take into account the increased cord and depth of the center section. A correction factor was derived by comparison of wing body combination To wing wing alone as carried out in N.A.C.A. Report No. 540.

Since combination 186 is comparable to the enclosed design, those C_D values were used in a comparative nature. First, the C_{De} differences were reduced on the ratio of .7 to 1 since the enclosed design has a center body approximately of that ratio in cross sectionly area to the body of combination of (186). Secondly, the new ratio of C_{De} of combination to wing alone was calculated for various angles of attack. The resultant value was noted on a percentage bases and appled to the enclosed design as a correction factor.

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1)

The C_D coefficient for nacelle drag and cooling drag was calculated as follows on a wing area bases.

(1.) $$C_D = \frac{391 \times D}{S \times V^2}$$

(2.) $$D = f \rho \frac{V^2}{2} \times N_m$$

C_D = Coef. based on wing area
S = Wing area
V = Vel. in m.p.h.
D = Drag in lbs.
f = Equivalent flat plat area. = 1.55
ρ = Density of air = .00237 @ Sea Level
N_m = No. of motors = 4

Combining (1.) + (2.)

$$C_D = \frac{391 \times f \rho \frac{V^2}{2} N_m}{S \times V^2}$$

$$= \frac{195.5 \times f \times \rho \times N_m}{S}$$

$$C_D = .122 \times f \times \rho \times N_m$$

The C_D coefficient for tail was calculated on bases of tail load, velocity and Reynold's no. and wing area. Results are tabulated as follows.

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C_D of ~~BODY~~ ELEVATOR

Velocity	80	99		127	156	221	256	313
R_e (Millions)	3	3.7		4.7	5.8	8.2	9.5	11.6
Tail Load	660	540		450	382	330	318	297
$\frac{\text{T. Load}}{.80}$	825	675		562	478	413	398	371
⊗ C_L	.315	.169		.0855	.0481	.0201	.015	.0093
C_{D_o} (CR_2 12)	.0077	.0078		.0078	.0078	.0073	.0072	.0070
$C_{D_o} + \frac{C_L^2}{\pi 10}$	.0108	.0087		.0080	.0078	.0073	.0072	.0070
C_D (Wing Bases)	.0010	.0008		.0008	.0008	.0007	.0007	.0007

⊗ These values are based on the C.G. 2" ahead and 10" below the a.c. on the M.A.C.

C_D of FIN

$C_{D\,min}$ R_e 8	N.A.C.A. .0009					.0064		
$C_{D\,min}$ (Wing Bases)	.00062	.00062		.00061	.00061	.00060	.00060	.00060

~ C_{D_5} of PILOTS WINDSHIELD ~

C_{D_1} of ⊂——— O = .94

C_{D_2} of ⊂——— D = .188 $\quad C_{D_1} - C_{D_2} = .752$

Assume Rear Half of Hemisphere (- Drag) = .752 × .3 = .226

C_{D_3} of ⊏——⊐ D $\quad .94 - .226 = .714$

C_{D_4} of End = .714 × .22 = .157 (Since 78% is Friction)

$C_{D_5} = .157 \times 6\square \times .78/1600 = .00046$

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2.

1	PERCENT of SEMI-SPAN		0	20	40	60	80	90	95	97.5	100
2	① × b/2 00	y	0								
3	Fig.	t									
4	Fig.	C	234	201	168.	136	103	86.5	78.4	74.35	70.25
5	③/④	t/c	.10	←						→	.10
6	Fig.	m_o	5.8	←						→	5.8
7	④ × ⑥	$m_o C$	1356	1166	975	788	597	502	455	432	408
8	⑥/m's	$m_o/m's$	1	←						→	1
9	Fig. 2-3 to	C'_{l_α}	.90	.985	1.03	1.07	1.05	.995	.87	.655	0
10	⑧ × ⑨	C_{l_α}	.90	.985	1.03	1.07	1.05	.995	.87	.655	0
11	④ × ⑩	CC_{l_α}	211	198	173	146	108	86	68	49	0
12		δ	.2	.8	.4	.8	.3125	.3375	.025	.10	.025
13	⑪ × ⑫	δCC_{l_α}	42.2	159	69	117	34	29	1.7	4.9	0
14	$C'_L = \frac{2}{S}\int_0^{b/2} CC_{l_\alpha}\, dy = \frac{b}{S}\frac{\Sigma(13)}{3} = \frac{1520}{230,000} \times \frac{456.8}{3} = 1.002$										
15	⑩/C'_L	$C_{l_{\alpha_1}}$	.90	.985	1.03	1.07	1.05	.995	.87	.655	0
16	1/m	α_a	.205	←						→	.205
17	⑮/⑥	α_o	.155	.170	.178	.185	.181	.171	.150	.113	0
18	⑯ - ⑰	W/V	.050	.035	.027	.020	.024	.034	.055	.092	-
19	⑮ × ⑱	C_{dia_1}	.045	.0345	.0278	.0217	.0252	.0338	.0478	.0603	-
20	⑲/C_{Dia_1}	$C_{d_{i_1}}$									
21	PERCENT OF AFFECTED TIP DISTANCE										
22	FIG 2-7	ΔC_{l_1}									
23	F_2F_3 × ㉒	$F_2F_3\,\Delta C_{l_1}$									
24	④ × ㉓	$F_2F_3\, C\Delta C_{l_1}$									
25	⑫ × ㉔	$F_2F_3\,\delta C\Delta C_{l_1}$									
26	$\Delta C_{L_1} = \frac{2}{S}\int_0^{b/2} F_2F_3\, C\Delta C_{l_1}\, dy = \frac{b}{S}\frac{\Sigma(25)}{3} =$										
27	FIG. 2-11	C.P.									
28	FIG.	a.c.									
29	FIG.	$C_{m_{a.c.}}$									
30		$C_{d_{o\ min(std)}}$	.0064	←						→	.0064
31											
32											

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1.5

$C_L = .15$ $C_D = .0064$ $L/D = 23.3$

$\alpha = 1.76°$

$V = 256$

1	PERCENT of SEMI-SPAN		0	20	40	60	80	90	95	97.5	100
2	Chord	C	19.5	16.77	14.04	11.31	8.58	7.22	6.54	6.19	5.85
67	R_e (Millions)		46.	39.6	33.2	26.8	20.3	17.0	15.5	14.6	
68		$C_{l\,max.}$	1.78	1.78	1.78	1.78	1.778	1.77	1.65	1.60	
69		$C_{l\,obt.}$	0.105	0.113	0.12	0.13	0.140	0.15	0.152	0.155	
70		$C_{l\,max} - C_{l\,obt.}$	1.675	1.667	1.66	1.65	1.638	1.62	1.498	1.445	
71		$C_{l\,ai}$	.90	.985	1.03	1.07	1.05	.995	.87	.655	
72		$C_{l\,max.}/C_{l\,ai}$									
73	.15 × $C_{l\,ai}$	C_l	0135	0.152	0.150	0.166	0.163	0.154	0.135	0.102	
74		$C_l - C_{l\,obt.}$	.030	.039	.04	.036	.023	.004	-.017	-.053	
75		$\frac{C_l - C_{l\,obt.}}{C_{l\,max} - C_{l\,obt.}}$	.0169	.022	.023	.020	.013	.0023	-.010	-.033	
76		ΔC_{do}	.00003	.00004	.00004	.00004	.00003	.00004	-.00002	-.00007	
77	$C_L^2 \times$ (19)	C_{di}	.00101	.00078	.00063	.00049	.00057	.00076	.00108	.00136	
78		$C_{do\,min}$	.00526	.00535	.00545	.00560	.00575	.00590	.00595	.00600	
79	(75)+(76)+(77)+(78)	C_d	.00630	.00617	.00612	.00613	.00635	.00670	.00705	.00743	
80		$C_d \times C$	.123	.104	.064	.094	.055	.048	.046	.046	
81	Σ (80)		0.580								
82	Σ (2)		90.15								
83	(81)/(82)	C_D (AV.)	.00643								

$C_L = .1$ $C_D = .0058$ $L/D = 17.2$

$\alpha = 1.175°$

$V = 313$

84	R_e (Millions)		56.	48	41	33	25	21	19	18
85	Assumed	ΔC_{do}	.00003	.00003	.00003	.00003	.00003	.00003	.00003	.00003
86	$C_L^2 \times$ (19)	C_{di}	.00045	.00035	.00028	.00022	.00025	.00034	.00048	.00060
87		$C_{do\,min}$	.00515	.00522	.00530	.00545	.00562	.00573	.00580	.00582
88	(85)+(86)+(87)	C_d	.00563	.00560	.00561	.00570	.00590	.00610	.00631	.00645
89		$C_d \times C$	.110	.094	.079	.065	.051	.044	.041	.040
90	Σ (89)		0.524							
91	Σ (2)		90.15							
92	(90)/(91)	C_D (AV.)	.0058							

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$C_L = .2 \quad C_D = .0072 \quad L/D = 27.8$

$\alpha = 2.35^\circ$

$V = 221$

1	PERCENT of SEMI-SPAN		0	20	40	60	80	90	95	97.5	100
2	Chord	C	19.5	16.77	14.04	11.31	8.58	7.22	6.54	6.19	5.85
93	R_e (Millions)		39.8	34.2	28.7	23.1	17.5	14.7	13.3	12.6	
94		$c_{l\,max}$	1.78	1.78	1.78	1.78	1.77	1.76	1.75	1.74	
95		$c_{l\,opt.}$	0.11	0.12	0.13	0.135	0.15	0.154	0.16	0.163	
96		$c_{l\,max} - c_{l\,opt}$	1.67	1.66	1.65	1.645	1.62	1.606	1.59	1.577	
97		$c_{l\,ai}$	.90	.985	1.03	1.07	1.05	.995	.87	.655	
98		$c_{l\,max}/c_{l\,ai}$									
99	$.2 \times c_{l\,ai}$	c_l	.18	.1970	.206	.214	.210	.1990	.174	.1310	
100		$c_l - c_{l\,opt.}$	.07	.077	.076	.079	.06	.045	.014	.068	
101		$\frac{c_l - c_{l\,opt.}}{c_{l\,max} - c_{l\,opt.}}$	.042	.046	.046	.048	.037	.028	.009	.043	
102		ΔC_{do}	.00010	.00010	.00010	.00010	.00008	.00006	.00002	.00010	
103	$C_L^2 \times$ (19)	C_{di}	.00180	.001380	.00111	.00087	.00101	.00135	.00191	.00241	
104		$C_{do\,min}$	.00533	.00543	.00554	.00570	.00584	.00596	.00600	.00640	
105	(103)+(104)+(105)	C_d	.00723	.00691	.00675	.00667	.00693	.00737	.00793	.00891	
106		$C_d \times C$	.141	.116	.095	.075	.060	.053	.052	.055	
107	Σ (106)		0.648								
108	Σ (2)		90.15								
109	(107)/(108)	C_D (Av.)	.0072								

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1.

$C_L = .4 \quad C_D = .0121 \quad L/D = 33.1$

$\alpha = 4.7^\circ$

$V = 156$ m.p.h.

1	PERCENT of SEMI-SPAN		0	20	40	60	80	90	95	97.5	100
2	Chord'	c	19.5	16.77	14.04	11.31	8.58	7.22	6.54	6.19	5.85
33	R_e (Millions)		28.1	24.2	20.3	16.3	12.4	10.4	9.4	8.9	8.4
34		$C_{l\,max.}$	1.78	1.78	1.77	1.775	1.74	1.72	1.70	1.69	1.685
35		$C_{l\,obt.}$	.13	.135	.141	.15	.161	.17	.174	.178	.179
36	$C_{l\,max} - C_{l\,obt.}$		1.65	1.645	1.629	1.625	1.579	1.55	1.526	1.512	1.506
37		$C_{l\,ai}$	.90	.985	1.03	1.07	1.05	.995	.87	.655	
38	$C_{l\,max} / C_{l\,ai}$										
39	$.4 \times C_{l\,ai}$	$= C_{l_2}$	.236	.394	.412	.428	.42	.398	.348	.262	
40	$C_{l_2} - C_{l\,obt}$		.23	.259	.271	.278	.259	.228	.174	.084	
41		$\frac{C_{l_2} - C_{l\,obt}}{C_{l\,max} - C_{l\,obt}}$	.14	.158	.166	.171	.164+	.147	.114	.056	
42	F	$\Delta C_{d\,o}$	.00044	.00053	.00057	.0006	.00056	.00047	.00033	.00013	
43	$C_l^2 \times$ (19)	c_{d_i}	.00720	.00552	.00445	.00347	.00402	.00540	.00765	.00965	
44		$C_{do\,min}$	.00556	.00565	.00575	.00590	.00610	.00620	.00630	.00633	
45	(42)+(43)+(44)	C_d	.01320	.01170	.01077	.00997	.01068	.01207	.01428	.01611	
46		$C_d \times C$	.257	.196	.151	.113	.0917	.0872	.0935	.0997	
47	$\leq$ (46)		1.089								
48	$\leq$ (2)		90.15								
49	(47)/(48)	C_D (AV.)	.0121								

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1

$C_L = .6$ $C_D = .0202$ $L/D = 29.7$

$\alpha = 7.5°$

$V = 127.5$ m.p.h.

1	PERCENT of SEMI-SPAN		0	20	40	60	80	90	95	97.5	100
2	CHord'	C	19.5	16.77	14.04	11.31	8.58	7.22	6.54	6.19	5.85
50	R_e (Millions)		23.	19.8	16.6	13.3	10.1	8.5	7.7	7.3	6.9
51		$C_{l\,max.}$	1.78	1.775	1.765	1.75	1.72	1.69	1.67	1.66	
52		$C_{l\,opt.}$	.135	.142	.150	.160	.170	.180	.183	.185	
53		$C_{l\,max} - C_{l\,opt.}$	1.645	1.633	1.615	1.585	1.550	1.51	1.487	1.475	
54		C_{2ai}	.90	.985	1.03	1.07	1.05	.995	.87	.655	
55		$C_{l\,max}/C_{2ai}$									
56	$C_L \times C_{2ai}$	C_l	0.54	0.59	.618	.642	.630	.597	.523	.394	
57		$C_l - C_{l\,opt}$	.405	.448	.468	.482	.460	.417	.340	.209	
58		$C_l - C_{l\,opt.}$ / $C_{l\,max} - C_{l\,opt.}$	.246	.275	.290	.304	.297	.276	.229	.142	
59		ΔC_{do}	.00111	.00138	.00153	.00168	.00159	.00138	.00098	.00044	
60	$C_L^2 \times$ (19)	C_{di}	.01620	.01243	.01000	.00780	.00906	.01216	.01721	.02170	
61		$C_{do\,min}$	.00570	.00580	.00590	.00600	.00625	.00635	.00640	.00650	
62		C_d	.02301	.01961	.01743	.01548	.01690	.01989	.02459	.02864	
63		$C_d \times C$	.449	.328	.245	.175	.144	.144	.161	.178	
64	Σ (63)					1.824					
65	Σ (2)					90.13					
66	(64)/(65)	C_D (AV.)				.0202					

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1. $C_L = 1.0 \quad C_D = .047 \quad L/D = 21.3$

$\alpha = 11.8°$

$V = 99$

1	PERCENT of SEMI-SPAN		0	20	40	60	80	90	95	97.5	100
2	Chord	C	19.5	16.77	14.04	11.31	8.58	7.22	6.54	6.19	5.85
110	Re (Millions)		17.8	15.3	12.8	10.3	7.82	6.58	5.97	5.65	
111		C_l max.	1.77	1.76	1.755	1.715	1.67	1.63	1.61	1.59	
112		C_l obt.	0.15	0.153	0.160	0.170	0.182	0.19	0.192	0.196	
113		$C_{l\,max} - C_{l\,obt.}$	1.62	1.607	1.595	1.545	1.488	1.44	1.418	1.394	
114		Cl_{ai}	.90	.985	1.03	1.07	1.05	.995	.97	.655	
115											
116	$C_L \times Cl_{ai}$	C_l	.90	.985	1.03	1.07	1.05	.995	.97	.655	
117		$C_l - C_{l\,obt}$	.75	.832	0.87	.90	0.868	.805	.678	.459	
118		$\frac{C_l - C_{l\,obt.}}{C_{l\,max} - C_{l\,obt}}$	.463	.418	.545	.582	.584	.559	.478	.330	
119		ΔC_{d0}	.00400	.00320	.00570	.00660	.00670	.00600	.00430	.00197	
120	$C_L^2 \times$ (19)	C_{di}	.04500	.03450	.02780	.02170	.02520	.03380	.04780	.06030	
121		$C_{d0\,min}$	.00585	.00595	.00605	.00620	.00640	.00655	.00660	.00663	
122		C_d	.05485	.04365	.03955	.03450	.03830	.04635	.05870	.06890	
123		$C_d \times C$	1.07	.732	.557	.390	.329	.334	.384	.433	
124	Σ (123)		4.229								
125	Σ (2)		90.15								
126	124/125	C_D (AV.)	.047								

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$C_{L\,max} = 1.54$ $C_D = .0155$ $L/D = 13.3$

$\alpha = 18.1°$

$V = 79.5$

1	PERCENT of SEMI-SPAN		0	20	40	60	80	90	95	97.5	100
2	Chord	C	19.5	16.77	14.04	11.31	8.58	7.22	6.54	6.19	5.85
127	R_e (Millions)		14.4	12.4	10.4	8.4	6.4	5.4	4.8	4.7	
128		$C_{z\,max}$	1.76	1.74	1.716	1.685	1.62	1.580	1.53	1.52	
129		$C_{z\,opt.}$	0.155	0.16	0.170	0.178	0.19	.200	.203	.204	
130		$C_{z\,max} - C_{z\,opt}$	1.415	1.58	1.546	1.506	1.43	1.380	1.327	1.316	
131		$C_{z\,ai}$	.90	.985	1.03	1.07	1.05	.995	.87	.655	
132		$C_{z\,max}/C_{z\,ai}$	1.95	1.76	1.67	1.57	1.54	1.58	1.76	2.32	
133	$C_L \times C_{z\,ai}$	C_z	1.26	1.520	1.585	1.648	1.62	1.532	1.34	1.010	
134		$C_z - C_{z\,opt}$	1.230	1.36	1.415	1.470	1.43	1.332	1.137	0.806	
135		$\frac{C_z - C_{z\,opt}}{C_{z\,max} - C_{z\,opt}}$	.868	.860	.915	.975	1.00	.965	.855	.613	
136		ΔC_{d_0}	.01800	.01750	.0225	.0335	.050	.031	.017	.0075	
137	$C_L^2 \times$ (19)	C_{di}	.10680	.09160	.0660	.0515	.0598	.0802	.1135	.1430	
138		$C_{d_{o\,min}}$	.00600	.00610	.0062	.0064	.0066	.0067	.0068	.0068	
139		C_d	.13080	.10520	.0947	.0914	.1164	.1179	.1373	.1573	
140		$C_d \times C$	2.55	1.764	1.330	1.033	1.000	.850	.895	.975	
141	Σ (140)					10.397					
142	Σ (2)					90.15					
143	141/142	C_D (AV.)				.0155					

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.5

ITEM									
									350
C_L	1.54	1.00	0.80	0.60	0.40	0.20	0.15	0.10	.08
α°	18.1	11.8		7.5	4.7	2.35	1.76	1.175	
WING - ASPECT=10 TAPER = .3 C_D	.1155	.047		.0202	.0121	.0072	.00643	.0058	
WING CORRECTION FACTOR	1.40	1.27		1.204	1.173	1.16	1.157	1.156	
WING FINALE C_D	.1620	.0596		.0244	.0142	.00835	.00745	.0067	
COOLING C_D	.0018	.0018		.0018	.0018	.0018	.0018	.0018	
TAIL 160 [illegible]	.0010	.00087		.0008	.00078	.00073	.00072	.0007	
FIN 150 [illegible]	.00062	.00062		.00061	.00061	.00060	.00060	.0006	
PILOTS WINDSHIELD	.00046	.00046		.00046	.00046	.00046	.00046	.00046	
COMPLETE AIRPLANE C_D	.16588	.06335		.02807	.01785	.01194	.01111	.01027	
L/D				21.5	22.6	16.8	13.7	9.9	

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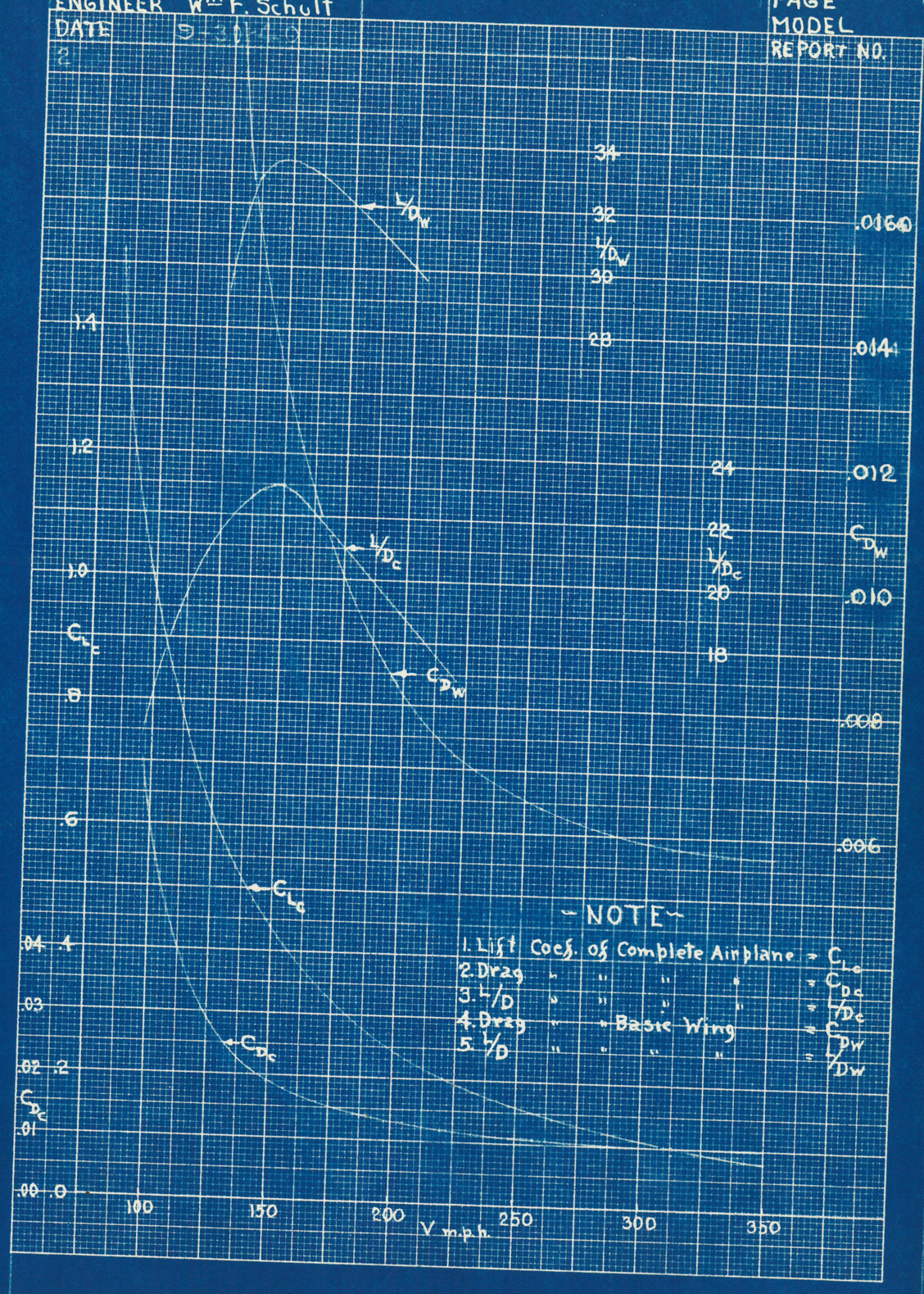

~NOTE~

1. Lift Coefs. of Complete Airplane = C_{L_c}
2. Drag " " " " = C_{D_c}
3. L/D " " " " = L/D_c
4. Drag " " Basic Wing = C_{D_W}
5. L/D " " " " = L/D_W

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1.

1. Design Altitude (Critical Altitude of Motor) = 14,500'
2. " Velocity V_d = 310
3. " Prop. r.p.m. N_d = 1350
4. " Brake H.P. (4 x 1000) $H.P._d$ = 4000
5. Number of Blades B = 3
6. Relative Density of Design Altitude σ_d = .6394
7. Pitch Limits
8. Propeller Dia. = 11'-0"
9. No. of Engines E = 4
10. C_D minimum = .0101
11. Wing Area S = 1600 □
12. Gross Weight W_g = 40,000 #
13. Aspect Ratio = 10

$$V_d = 50\sqrt[3]{\frac{1000 \times 4}{1600 \times .0101 \times .6394}} = 365 \quad \sigma = 310 \times 1.25 = 386$$

$$C_s = \frac{.638 \times 386 \times .925}{3.98 \times R.P.M.^{2/5}} = \frac{57.2}{R.P.M.^{2/5}}$$

$$Dia = \frac{386 \times 88}{R.P.M \times \frac{V}{ND}} = \frac{3400}{R.P.M. \times \frac{V}{ND}}$$

$$C_p = \frac{1000}{2 \times .6394 \times \left(\frac{R.P.M}{100}\right)^3 \times \left(\frac{Dia}{10}\right)^5} = \frac{784}{\left(\frac{N}{100}\right)^3 \left(\frac{D}{10}\right)^5}$$

Prop. r.p.m. =	1350	1800	2000
	= 3.19	2.85	2.73
$\frac{V}{ND}$ =	2.3	1.85	1.75
Dia =	10.95'	10.25'	9.75'
Vtip =	775	965	1020
C_p =	.203	.1185	.10
C_T =	.075	.057	.05
η =	85	89	87.5

Propeller used for Design 5868-9 N.A.C.A. Report 658 Fig. 14, 15, 16, 17
Dia. = 11'-0"

$$C_L = \frac{391 \times W}{S \times V^2} = \frac{9800}{V^2}$$

$$P_r = \frac{C_D \times S \times V^3}{146600} = .0109 \times C_D \times V^3 \times \sqrt{\frac{1}{\sigma}}$$

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σ = .639, $\sqrt{\frac{1}{\sigma}}$ = 1.25 — 14500′ — C_p = .203 — N = 1350 — H.P. = 4000

$V_{s.l.}$	C_L	C_D	Pr s.l.	Pr. a.	V_a.	$\frac{V}{ND}_a$	C_T	η%	V_{Tip}	$\eta_\oplus$	$\eta_\ominus$	H.P.$_{RV}$	H.P.$_e$
284		.0103	2570	3210	355	2.10	.085	88	953	.95	84	3350	140
270		.0105	2250	2810	338	2.00	.09	88.6			84	3360	550
225		.0115	1430	1786	281	1.67	.106	87.2			83	3320	1534
180		.0149	948	1185	225	1.33	.125	82.5			79	3160	1975
120		.0330	621	777	150	0.89	.152	66.6			63	2520	1743
152.		.0190	728	910	190	1.13	.141	79		.95	75	3000	2090
80					100								

σ = 1 — $\sqrt{\frac{1}{\sigma}}$ = 1. — SEA LEVEL — C_p = .130 — N = 1350 — H.P. = 4000

300		.0102	3010			1.78	.065	89	890	.95	.85	3400	390
270		.0105	2250			1.61	.071	88		.95	.84	3370	1120
225			1430			1.34	.085	88		.95	.84	3370	1940
180			948			1.07	.100	82.5		.95	.79	3160	2212
152			728			.905	.115	80.0		.95	.76	3040	2312
120			621			.714	.132	72.5		.95	.69	2760	2139
80			930			.473	.15	54.5		.91	.518	1035	

σ = .8616 — $\sqrt{\frac{1}{\sigma}}$ = 1.077 — 5000′ — C_p = .151 — N = 1350 — H.P. = 4000

300		3010	3240	323	1.91	.070	88.5		.95	84	3360	120
270		2250	2420	291	1.72	.077	87.5			83	3320	900
225		1430	1540	242	1.43	.091	86.0			82	3290	1750
180		948	1020	194	1.15	.110	83.3			79	3170	2150
152		728	785	164	0.97	.125	80.1			76	3040	2255
120		621	670	130	0.77	.141	71.7		.95	68	2720	2050

σ = .7384 — $\sqrt{\frac{1}{\sigma}}$ = 1.1637 — 10000′ — C_p = .176 — N = 1350 — H.P. = 4000

300			3010	3500	349	2.07	.074	87.		.95	82.6	3310	190
270			2250	2610	314	1.86	.082	86.5			82.2	3290	680
225			1430	1660	261	1.54	.098	86.0			81.6	3270	1610
180			948	1100	209	1.24	.116	81.5			77.5	3100	2000
152			728	847	176	1.05	.132	78.5			74.7	2990	2143
120			621	722	139	0.82	.149	69.5		.95	66.0	2650	1928

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$\sigma^{1.5}$ = .5327 $\sqrt{\frac{1}{\sigma}}$ = 1.37 — 20000′ — C_p = .244 — N = 1350 — H.P. = 3320

$V_{S.L.}$	C_L	C_D	$P_{r.S.L.}$	$P_{r.2.}$	V_R	$\frac{V}{N.D.}$ a	C_T	$\eta\%$	$V_{T.P.}$	η_Φ	η	H.P.av.	H.P.e
259		.0108	1970	2700	354	2.10	.101	86.8		.95	82.5	2740	40
233		.0115	1590	2180	321	1.90	.111	86.5			82.3	2730	550
197		.0130	1082	1482	270	1.60	.126	82.5			78.3	2600	1118
161		.0175	795	1090	220	1.30	.141	75.2			71.5	2370	1280
136		.0235	645	885	186	1.10	.146	65.8			62.5	2080	1195
98		.0680	642	880	135	.80	.156	51.2		.95	48.7	1620	740

σ = .4480 $\sqrt{\frac{1}{\sigma}}$ = 1.494 — 25000′ — C_p = .290 — N = 1350 — H.P. = 2800

226		.0119	1500	2240	338	2.00	.125	86.2		.96	82	2300	60
203		.013	1190	1780	304	1.80	.134	83.0			79	2210	430
169		.016	839	1252	253	1.50	.148	76.5			73	2050	798
136		.024	659	983	203	1.20	.155	64.2			61	1710	727
113		.039	542	810	169	1.00	.157	54.2		.96	52	1460	650
					135	0.80							

σ = .374 $\sqrt{\frac{1}{\sigma}}$ = 1.635 — 30000′ — C_p = .348 — N = 1350 — H.P. = 2280

196		.0134	1100	1800	321	1.90	.147	80.2		.95	76	1736	-64
175		.0152	885	1450	287	1.70	.151	73.6			70	1597	147
144		.0208	677	1106	236	1.40	.158	63.4			60	1370	264
124		.03	625	1020	203	1.20	.162	55.7		.95	53	1210	190

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EXCESS H.P. for CLIMB

GROSS WT. = 40,000#

$R/C = H.P._e \times .825$

SEA LEVEL
5000'
10000'
14500'
20000'
25000'
30000'

H.P.$_e$

2000
1800
1600
1400
1200
1000
800
600
400
200
0

100 150 200 250 300 350

Vel. in M.P.H.

KEUFFEL & ESSER CO., N. Y. NO. 359-14
Millimeter, 10th lines heavy.
MADE IN U.S.A.

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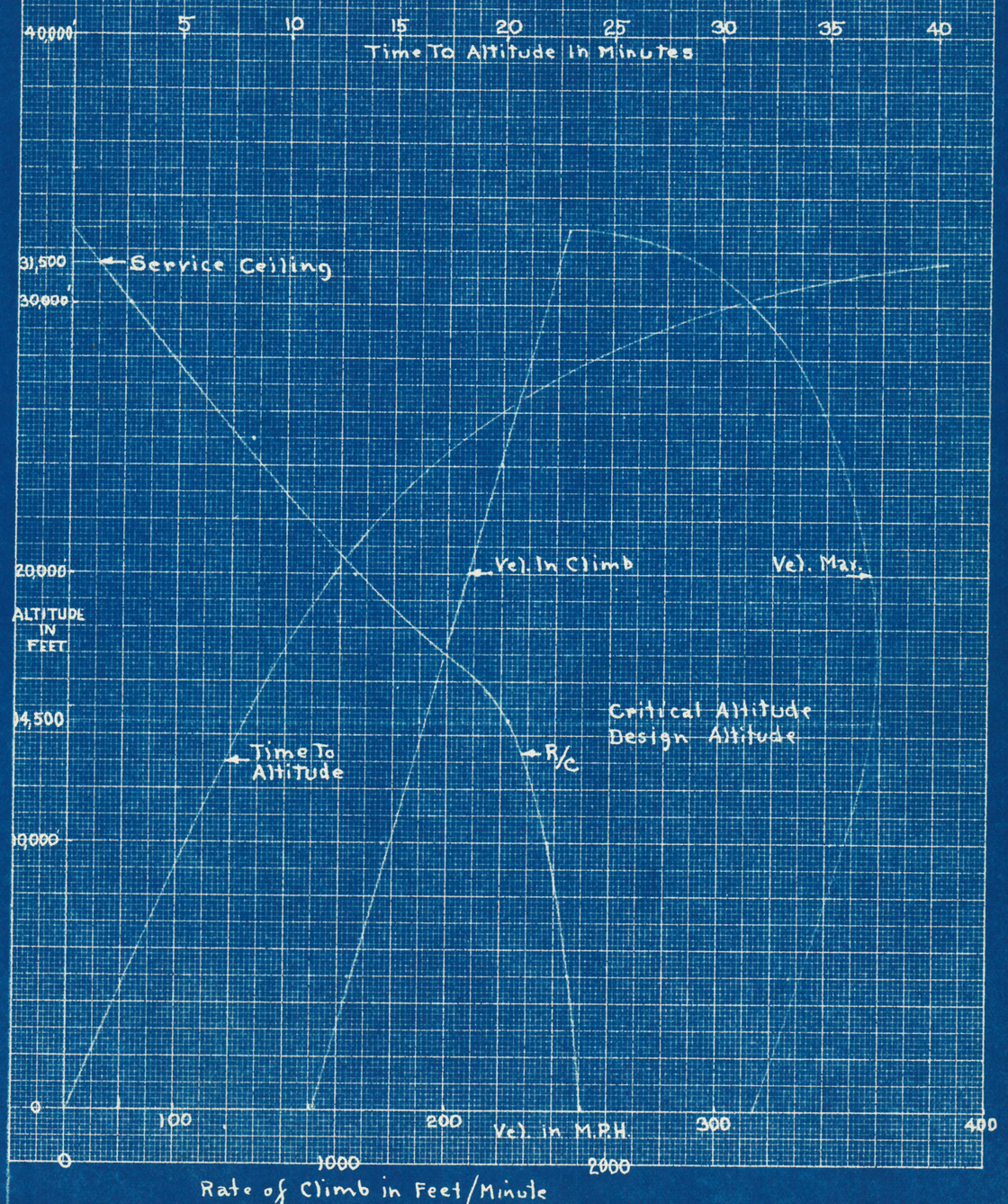
PERFORMANCE @ ALTITUDE

4-1000 H.P. P.+W. MOTORS — GROSS WT. - 40000*



KEUFFEL & ESSER CO., N. Y. NO. 359-14
Millimeters, 11th lines heavy.
MADE IN U.S.A.

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DESIGN LOAD = 40,000# WING AREA = 1600

RANGE @ HIGH SPEED @ DESIGN ALTITUDE of 14,500'
with FULL LOAD

$R_{vm} = \frac{\text{Fuel (max) in Gal's} \times V_{max}}{\text{Fuel Consumption in Gals/Hp @ } V_{max.}}$

Assume 5 gals x 4 = 20 gallons is used to warm up motors
At take off 400/60 or 6.66 gal's are used per minute
Time to climb To 14,500' altitude with full load = 8.1 minutes
Fuel used in climb = 6.66 x 8.1 = 54 gals
Total fuel at 14,500' = 800 - 20 - 54 = 726 gals

$R_{vm} = \frac{726 \times 357}{400} = 648$ miles = 648 miles

$R_B = 863 \times \frac{\eta_{max.}}{C_{min}} \times \left(\frac{L}{D}\right)_{max} \times \text{Log}_{10}\left(\frac{W_{gross}}{We}\right)$

$\eta_{max.} = .84$

$C_{min.} = .45$ #/B.H.P./hp.

Fuel Used = 4800 #
Oil " = 660

$\text{Log}_{10}\left(\frac{We}{We}\right) = \frac{40000}{35200} = 1.14 = .0569$

$L/D_{(max)} = 23$

$R_B = 863 \times \frac{.84}{.45} \times 23 \times .0569 = 2110$

$R_{max.} = .80(R_B - R_{vm}) + R_{vm}$
$= (.80 \times 1462) + 648 = 1818$ $R_{max.} = 1818$ miles

$V_R = \sqrt{\frac{195\,(W_G + We)}{\sigma\, S\, C_{L.R.}}}$

$\sigma = .6394$

$S = 1600$

$C_{L.R.} = .45$

$V_R = \sqrt{\frac{195 \times 75{,}200}{.6394 \times 1600 \times .45}} = 178$ m.p.h. $V_R = 178$ m.p.h.

$R_{1000} \sim F_r = \frac{1000 - 648}{2110 - 648} = \frac{352}{1462} = .241$ $F_V = .20$

$V_{1000} = 357 - .2(357 - 178) = 357 - 35.8 = 321.2$ $V_{1000} = 321$ m.p.h.

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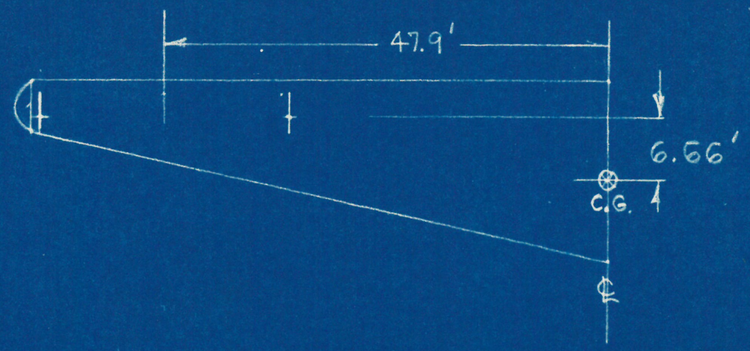
1.5

- RUDDER CONTROL -



Rudder Area (2 × 37.5) = 75 □'
Section = N.A.C.A. 0009
Rudder Rotation in α = 20° 17.1
Force Coeff. C_L = 1.02 1.20 ($C_{L\,max}$)
C_D = .46 0.10

Moment About C.G. = $\frac{C \times 75 \times V^2}{391} \times$ Dist.

$M_{C.G.} = (.192 \times C_L \times \text{Dist.} + .192 \times C_D \times \text{Dist.}) V^2$
$= .192 V^2 (C_L \times \text{Dist} + C_D \times \text{Dist})$

N_c (Yawing Moment Coef) = $\frac{\text{Moment}}{\text{Span} \times \text{Area of Wing} \times V^2}$

$N_c = \frac{.192 V^2 (C_L \times \text{Dist.} + C_D \times \text{Dist.})}{126 \times 1600 \times V^2}$

$N_c = .000000954 (C_L \times \text{Dist.} + C_D \times \text{Dist})$

When $\alpha = 20°$

$N_c = .000000954 (1.02 \times 6.66 + .46 \times 47.9)$
$= .000000954 (6.8 + 22.1) = .0000275$

When $\alpha = 17.1$ ($C_{L\,max}$)
$N_c = .000000954 (8 + 4.8) = .0000122$

Note: For Good Rudder Control N_c = .000015 - .000025

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⊛ HORIZONTAL TAIL AREA

$W = 40{,}000$ #

$S_w = 1600$ □

$\frac{W}{S} = 25$ #/□

Section = N.A.C.A. - 24 (.10 Thickness)

Aspect $R_w = 10$

$\ell = 323$

$C = 167$

$\frac{\ell}{C} = 1.93$

Aspect $R_t = 8$

$\frac{dC_L}{d\alpha}$ (Wing) A.R. = 10 F_4 = .085

$\frac{dC_L}{d\alpha_t}$ (Tail) A.R. = 8 F_1 = .081

Downwash Factor $\left\{\begin{matrix} A.R_w = 10 \\ \frac{\ell}{C} = 1.93 \end{matrix}\right\}$ F_2 = .70

Wing Section Stability Factor F_3 = .25

Stability Coefficient K = .0006

C.G. ~ $\frac{a}{C}$ = .20, .22, .24

$$\frac{S_t}{S_w}\frac{\ell}{C} = \frac{1}{F_1 F_2}\left[K\left(\frac{W}{S_w}\right) + \left(\frac{a}{C} - F_3\right)F_4\right]$$

$$S_t = 1250\left(-.103 + \frac{a}{C}\right)$$

$\frac{a}{C}$ =	.20	.22	.23	.24	.26
S_t □ =	122	146	159	172	197

⊛ This calculation is based on N.A.C.A. Report #293

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~ WING CHARACTERISTICS ~

Airfoil Sect.=N.A.C.A. 24(10) Area=1600 □", E.M.A.R.=10, Incidence=0

x=-9 z=+32

ITEM									
C_L	1.54	1.00	.60	.40	.20	.15	.10	.08	.07
a_o deg.	18.1	11.8	7.5	4.7	2.35	1.76	1.175	.94	.82
C_{mo}	0	0	0	0	0	0	0	0	0
C_{DW}	.1155	.047	.0202	.0121	.0072	.00643	.0058	.0057	.0058
Δa deg.=18.24 $C_L/E.M.A.R.$	2.81	1.83	1.095	.73	.365	.274	.183	.146	.128
a_w deg.= $a_o + \Delta a$	20.91	13.63	8.59	5.43	3.71	2.03	1.358	1.086	.948
θ deg.									
$C_L \cos\theta$	1.44	.9718	.592	.398			.0999		.07
$C_{DW} \sin\theta$	.0413	.0111	.0030	.001145			.000136		.000096
$C_L \sin\theta$	.55	.2355	.0900	.00378			.00235		.001162
$C_{DW} \cos\theta$	.108	.0456	.0200	.0120			.00579		.0058
$C_z = C_L \cos\theta + C_{DW} \sin\theta$	1.48	.9829	.595	.3991			.10576		.07009
$C_x = C_{DW} \cos\theta - C_L \sin\theta$	-.442	-.1899	-.07	+.0082			+.00344		+.00464
$C_{mz} = C_z \times \frac{x}{m.a.c}$	-.0797	-.053	-.0321	-.0215			-.00571		-.00378
$C_{mx} = C_x \times \frac{z}{m.a.c.}$	-.0846	-.0363	-.0134	+.00157			+.00066		+.00089
$C_{mw} = C_{mo} + C_{mz} + C_{mx}$	-.1643	-.0893	-.0455	-.01993			-.00637		-.00289
$C_t = -C_{mw} \frac{m.a.c}{d_t}$									

$\frac{x}{m.a.c} = \frac{-9}{167} = -.054$ $\frac{z}{m.a.c} = \frac{32}{167} = +.1915$

location of m.a.c. on M.A.C.

From tests = .25 × 167

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- TAIL CHARACTERISTICS -

Wing Area S_w = 1600 □"
Tail Area S_t = 160 □"
Span of tail b_t = 470 "
Chord of tail (Max.) C_x = 74 "
Distance (C.G. of plane to C.P.$_t$ = 323 "
Gross weight of Airplane = 40,000 #
Aspect ratio of tail = 10
$F_{a.r.}$ = 1.10

$$\frac{\Delta C_{Lt}}{\Delta a_t} = .072 \times F_{a.r} = 0.092$$

Air speed reduction on tail f_v = 0.81

$$C = \frac{S_t \times \frac{\Delta C_{Lt}}{\Delta a_t} \times f_v \times d_t}{S_w \times m.a.c.}$$

$$= \frac{160 \times .092 \times .81 \times 27}{1600 \times 13.92} = -.0145$$

Wing incidence i_w = 0
Tail " i_t = -.5
Downwash = $\epsilon = \frac{C_L}{.144\ F_{A.R.W}}$ = 6.32 C_L
C_{mt} = -.0145 a_t

C_{Lw}	1.54	1.00	.60	.40	.20	.15	.10	.08	.07
a_w	20.91	13.63	8.59	5.43	3.71	2.03	1.36	1.09	.95
C_{mw}	-.164	-.089	-.0455	-.020			-.0064		-.0029
θ deg.	20.91	13.63	8.59	5.43	3.71	2.03	1.36	1.09	.95
ϵ deg	9.8	6.32	3.89	2.53	1.26	0.95	0.86	0.69	0.60
a_t	10.6	6.81	4.20	2.40	1.95	.58	0.003	-0.10	-0.15
C_{mt}	-.153	-.098	-.061	-.035	-.0383	-.0081	.0	+.0015	+.0022
C_m (Final)	-.317	-.187	-.107	-.055			-.0064		-.0007
							?		?

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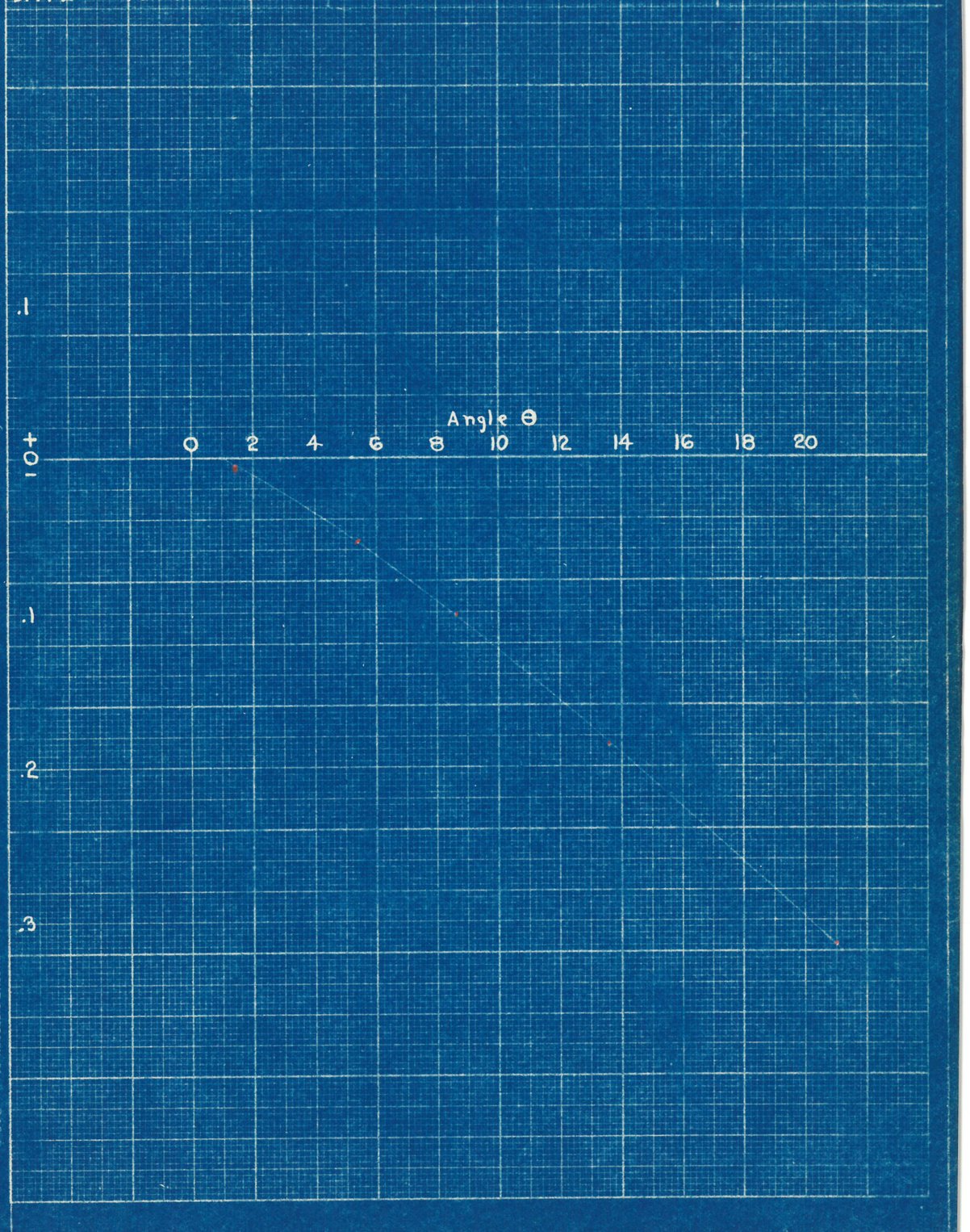

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